Technical Memorandum No. 33-42

A PRELIMINARY STUDY OF ADVANCED PROPULSION SPACECRAFT PAYLOAD CAPABILITIES

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CONTENTS

FIGURES

I. INTRODUCTION

This Memorandum outlines a preliminary study which has been made to illustrate the uses and potentials of electrical propulsion. Results include performance values for several electrical propulsion spacecraft missions.

The performance of a system for a given mission is presented in terms of gross payload (defined as terminal mass minus powerplant mass) as a function of flight time. The gross payload includes structures and tankage, scientific instrumentation, guidance and control weight, instrumentation and telecommunication equipment, and the ion motor itself. In addition to the gross payload, the power source will also be available, on arrival, for communication purposes. The flight time shown is not necessarily the same as propulsion time; most missions will be flown at a constant power-on thrust level and will require a considerable power-off coast time.

II. GENERAL ASSUMPTIONS

In order to make this study, estimates were made of the efficiency of an electrostatic ion motor. An estimated curve of the variation of ion motor efficiency versus specific impulse is shown in Fig. 1. The lower curve, labelled Motor 1, represents a conservative set of values; these efficiencies should be obtainable within the next year. Most of the calculations in this Memorandum are based on what is here termed Motor 2, a performance level which should be available in the next 4 to 5 years. However, it should be noted that, for ion rockets, the range below about 5000 sec may be considerably more difficult to achieve than that above 5000 sec; and therefore performance values at 2000-3000 sec should not really be considered as desirable for ion rockets. Improvements in MHD devices may eventually permit efficient operation in this range. The variation of thrust with efficiency is not limited to ion rockets, but is characteristic of any separately powered thrust device. The relationship between thrust, efficiency, and specific impulse is defined by the following formulas:

$$\eta \equiv \frac{F\, I_{sp}}{45.85\, P_0} \text{ and } I_{sp} \equiv \frac{F}{\dot{M}}$$

where

F = thrust (lb)

$\dot{m}$ = mass flow rate (lb/sec)

P_0 = input power to thrust device (kw)

The behavior of thrust per unit power is shown in Fig. 2. Note that the thrust per unit power has a definite peak value. Operation at peak thrust will determine the shortest possible time for any mission, but it is generally desirable to operate at a somewhat higher I_{sp} than that producing peak thrust, if possible. Clearly there is no point in considering an I_{sp} less than that corresponding to peak thrust.

III. PRESENT STATE OF THE ART

The first available muclear electric spacecraft will probably be one with a SNAP-8 type power source, delivering 60 kw to the thrust unit and having a total weight of 3000 lb. This powerplant is taken as representative of the present state of the art, and is chiefly compatible with lunar, Mars, and Venus missions.

Figure 3 shows a lunar orbiter mission, performed at constant thrust, with approximately 5 days of motor-off coast time near the end of the mission. The trajectory for this mission is described in Ref. 1. The spacecraft is assumed to be

Centaur-boosted, giving an initial weight of 8800 lb in a 300 n. mi. Earth orbit. The 60-kw--Centaur booster combination is capable of putting a 3800-lb payload into a 95 n. mi. lunar orbit in 130 days, or 4600 lb in 200 days. Note that these times are well within the nominal 10,000-hr lifetime of the SNAP-8 system, so that, in addition to this payload, a live reactor giving 60 kw of electric power is also available. Also note that even for a relatively low-energy mission such as this, going to specific impulses below 3000 sec means paying a fairly heavy payload penalty for a relatively small decrease in flight time. It appears that 4000 to 6000 sec is a desirable range of specific impulses even for lunar missions, using a nuclear-electric propulsion system. Electric thrust devices operating in the range 1000 to 2000 sec, such as arc jets, do not appear to be as desirable as either nuclear rockets (at 1000 sec) or ion rockets.

The second curve shown in Fig. 3 is that for a 30-kw, 2000-lb powerplant. At a given specific impulse, the flight time is nearly doubled in dropping from 60 to 30 kw, but an extra 1000 lb is available for payload since the 30-kw powerplant is lighter by that amount. Therefore, at 5000 sec, the flight time has increased from 180 days (with 60 kw) to 350 days, (with 30 kw), while the payload has gone from 4500 to 5500 lb. The lunar orbiter appears to be the limit of ion-motor mission feasibility with a 10,000-hr, 30-kw powerplant. Any interplanetary missions would take much longer than the powerplant lifetime.

All of the interplanetary missions to be described here start from a 300 n. mi. Earth orbit, and terminate in one of three ways: (1) flyby missions, which intercept the orbit of the destination planet but do not match velocities with it; (2) capture missions, which arrive at the destination planet at the same time and with the same velocity as the planet and will therefore achieve some kind of elliptical planetary orbit; and (3) orbiter missions, which terminate with the payload in a stable circular orbit at some desired altitude.

The missions analyzed consist of three phases: (1) a slow spiral out to escape from the initial Earth orbit, (2) a heliocentric transfer from Earth's orbit to the destination orbit, and (3) for orbiters, a slow spiral into the final planetary orbit. The treatment followed is basically that of Ref. 2.

The planetary spirals for escape and capture are performed with constant tangential thrust, which is very nearly optimum. The heliocentric transfer trajectories are generated by the so-called Irving and Blum method, an optimal, variable-thrust and variable-specific-impulse program. Both types of trajectories are fully described in Ref. 3. The optimizations referred to are those which minimize fuel consumption for a given total flight time. Varying specific impulse over a wide range is not a realistic way to operate an ion motor; but preliminary studies have shown that an interplanetary transfer flown with constant thrust and a coast period in the middle portion will only result in final payloads about 10% less than the optimal, variable-thrust program. In addition, the desirable single specific impulse for a constant thrust transfer will tend to be slightly higher than those indicated here.*

Figure 4 shows a Venus capture mission, for the 60-kw Centaur-boosted spacecraft. The payload capabilities for this mission are 1850 lb in 240 days or 4200 lb in 380 days. The specific impulses indicated are those used in the constant-thrust

*If efficiencies lower than those shown in Fig. 1 must be used, the effect will be to lengthen the powered flight time at a given specific impulse, leaving the payload unchanged.

Earth escape maneuver. For the interorbital transfer, it was assumed that the beam power, (ηP_0) remained the same as it had been during the escape phase. Payload estimate may be either raised or lowered by varying the initial parameters and trajectory assumptions; in general those given here are probably good to ±15%.

The envelope curve (solid line) represents the best performance attainable and shows where the optimal specific impulse for a given flight time will lie. This optimization is, as will be discussed later, extremely sensitive to the initial parameters chosen for the mission and therefore should not be taken as a firm number.

The optimal transfer trajectory can be computed for a very wide range of flight times. However, in order to fly the Irving and Blum trajectory, the vehicle must be able to follow a prescribed acceleration program for each value of transfer time. The magnitude of acceleration required is a maximum at the beginning and end of the transfer trajectory, and decreases nearly to zero in the middle portion. As discussed earlier (Fig. 2) the thrust obtainable with a given powerplant has a definite maximum value. There is thus a corresponding maximum acceleration, a_{max}, available at the beginning of the interplanetary transfer. The maximum acceleration required to transfer in a time T_o (called a_o) is given in Ref 3. It has been assumed in this study that the shortest transfer time for any mission is that which gives approximately $a_o = a_{max}$. This condition gives the left-hand cutoff of the curves in Fig. 4 (and the following figures) and determines the envelope curve. For constant thrust missions, the shortest mission time may be somewhat less (possibly 10%) but the payload will be dropping very steeply with mission time. It appears that assuming $a_o = a_{max}$ is a quite reasonable assumption for short flight times.

Figure 5 shows a Mars capture mission. The payload capabilities are 1500 lb in 300 days, or 3200 lb in 400 days; this mission is thus somewhat more difficult than a Venus capture. But for both Mars and Venus missions, the 60-kw reactor will still be available as a communication power source on arrival.

Figure 6 is a Mars orbiter mission, terminating in a 500 n. mi. orbit about Mars. The payload which can be carried on a 400-day trip is 2250 lb. For this mission, note that to arrive in 400 days, it appears necessary to fly a large part of the mission at a low specific impulse - much lower than would be desirable from an ion motor design point of view. In part, this requirement is due to the assumptions made in defining the mission. As noted earlier, flying the mission entirely at one thrust level tends to raise slightly the specific impulse for a given trip time. A more marked effect can be achieved by initiating the mission at a higher altitude than 300 n. mi. Figure 7 shows the same Mars orbiter as Fig. 6, with, for comparison purposes, a mission starting from an altitude of 1000 n. mi. and flown at constant thrust. Under these conditions it would be possible to operate at 4000-5000 sec to arrive in 400 days, with about a 2000-lb payload. The initial spacecraft weight at 1000 n. mi. has dropped to 7480 lb. But this is largely compensated for by the shorter escape time and the decrease in thrust required (for the same acceleration) and therefore the lower mass flow rate and higher efficiency for the nuclear-electric portion of the trip. The constant thrust trajectories are not optimized, but appear to be quite close to those for minimum fuel consumption, (Ref. 4). Choice of the best initial altitude for a mission will clearly be a function of the booster and the nuclear electric spacecraft system capabilities.

There are two other missions which may be performed with a Centaur-boosted SNAP 8 powerplant. One is a Jupiter flyby, arriving in 850 days with an 1800-lb payload, or in 630 days with a 900-lb payload. The latter flight will continue on to escape from the solar system. In addition, an out-of-the-ecliptic probe can be sent to an inclination of 15 deg with a payload of about 2000 lb.

IV. FUTURE STATE OF THE ART

The next step in powerplants will probably be a 300 kw to 1 mw, 10 lb/kw system. Considering this power source on a Saturn booster, the following missions become feasible.

Figure 8 shows a Mercury capture mission. The 300-kw powerplant will arrive with 14,000-lb payload in 300 days, or 20,000-lb in 400 days. The 1-mw system will arrive with 10,000 lb in 160 days, 18,000 lb in 200 days, or correspondingly higher payloads in longer flight times.

Figure 9 shows a Venus capture mission again. The 300-kw system arrives with 21,000 lb in 240 days, or 33,000 lb in 380 days. This is an order of magnitude increase in payload over the 60-kw system capability. For a Mars capture or orbiter, the payload will again be an order of magnitude greater with a 300-kw system than with a 60 kw system. The 1-mw system is probably more useful for decreasing mission time than for increasing payload.

Figure 10 is a Jupiter capture mission. For high energy missions such as this, the 1-mw power level is desirable, as is a high specific impulse such as 10,000 sec. The payload capability of the 1-mw system is 26,000 lb in 800 days or 16,000 lb in 550 days. A curve for the performance of a 10 mw, 10,000-lb powerplant is also included to indicate the power levels that will be needed for short (1 year) capture trips to the outer planets, although no such powerplant appears feasible in the near future.

Figure 11 is a Saturn capture mission, the highest energy mission considered so far. The 1 mw powerplant will arrive with a payload of 12,500 lb in 730 days, or 21,000 lb in 970 days. Again a 10 mw system is representative of desirable future power levels; it will arrive with payloads ranging from 4000 lb in 250 days to 26,000 lb in 550 days.

V. DIRECT NUCLEAR SYSTEMS

Direct nuclear systems have not been as extensively studied as electric systems; however a few preliminary results have been obtained for comparison purposes. Consider the same booster which has been discussed for electric propulsion; that is, initiate a mission with an initial weight of 45,000 lb in a 300 n. mi. Earth orbit but with a 1000 sec. specific impulse, high thrust, direct nuclear engine for interplanetary propulsion. (High thrust means that accelerations of more than 0.5 g are available; there will be no low thrust penalty of increased characteristic velocity required for a mission. In this case, a thrust of at least 25,000 lb would be required, corresponding to a 500-mw reactor).

Since there is much uncertainty about the weights associated with such a propulsion system, comparisons will be made on the basis of terminal mass only - that is, initial mass minus propellant mass.

For a Mars orbiter mission, a direct nuclear system can place 58% of the initial orbital mass (or 26,000 lb) in a 500 mi Martian orbit in 260 days. (Ref 5.) A

1-mw electric system can place 82% (or 36,000 lb) in this orbit in 320 days. The powerplant weight associated with each system must be subtracted from these figures to obtain the final payload; however the powerplant of the electric system is still available for communications and other power needs. It then appears that the two systems have quite comparable performance, in terms of terminal mass, for Mars missions. The flight time advantage lies with the nuclear system; the nuclear-electric system will arrive with a greater mass.

If we consider a higher energy mission, such as a Jupiter flight, the situation is somewhat different. Figure 12 shows the ratio of terminal mass to initial mass, as a function of flight time, for (1) a 1-mw electric system performing Jupiter flyby and Jupiter capture missions and (2) a 1000-sec, 500-mw direct nuclear system performing a Jupiter flyby mission. Here it is clear that the most desirable specific impulse is high; for a probe mission, the 10,000-sec electric system will arrive with 65% of its initial mass in 300 days, or with 87% in 600 days. The direct nuclear system at 1000 sec can arrive with at most 51% of its initial mass in 950 days. In fact, both time and weight advantages are still with the high specific impulse system for the more difficult Jupiter capture mission.

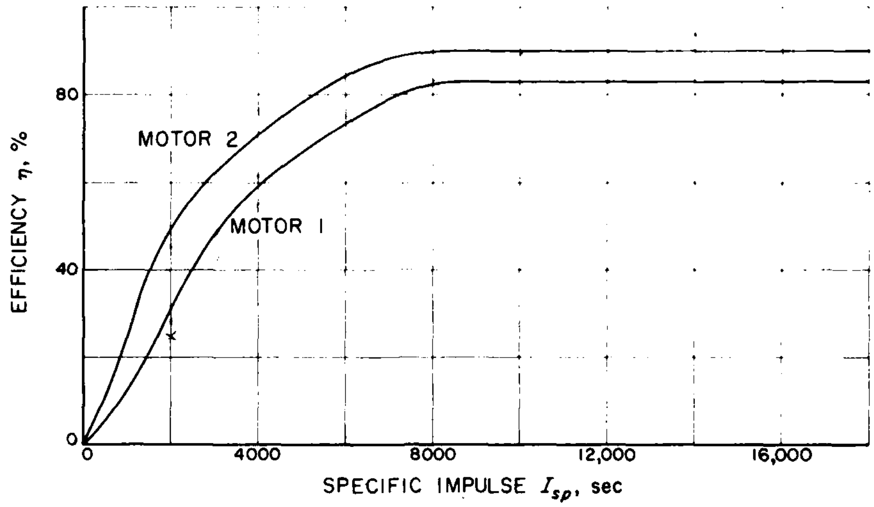

Fig. 1. Variation of thrust device efficiency with specific impulse

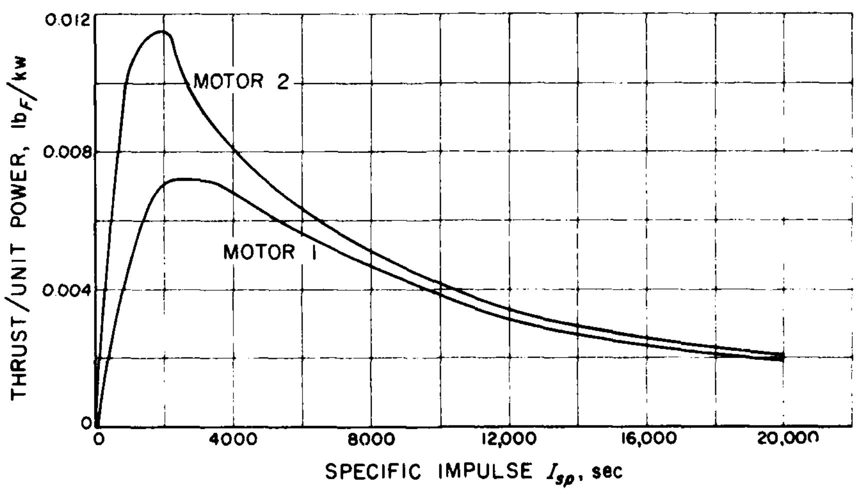

Fig. 2. Thrust per unit power versus specific impulse

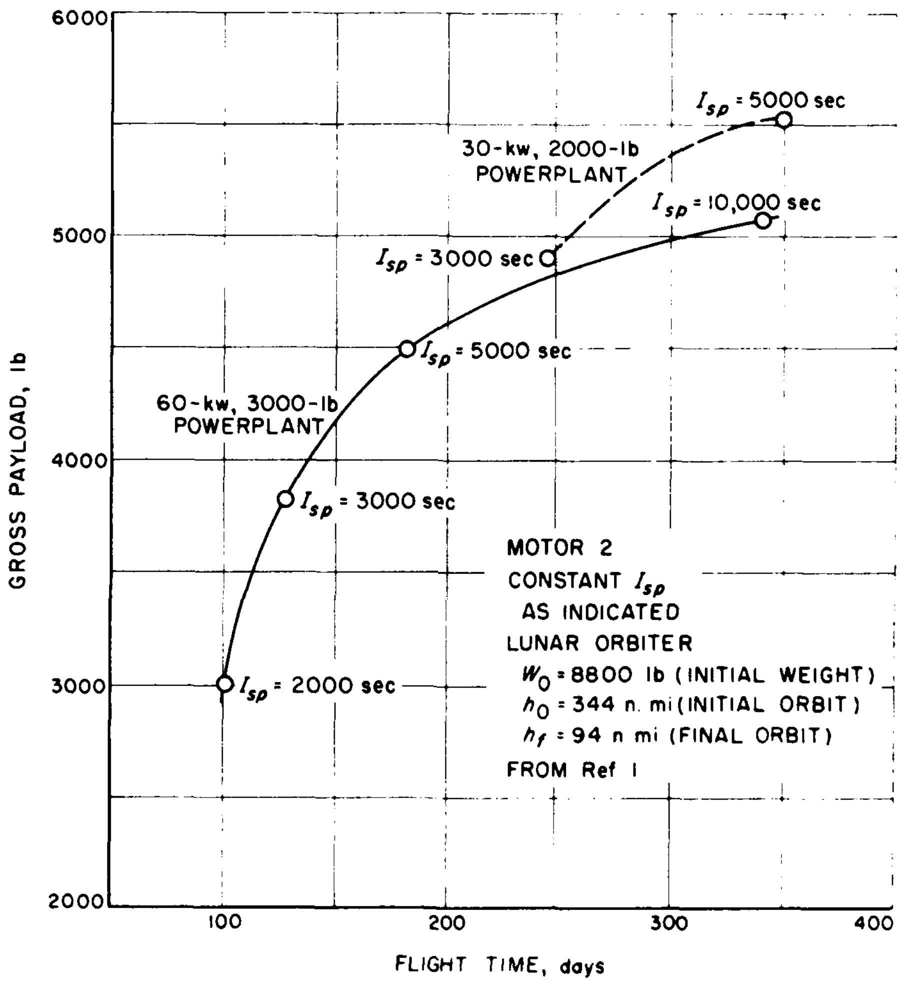

Fig. 3. Lunar Orbiter mission, 30 kw and 60 kw

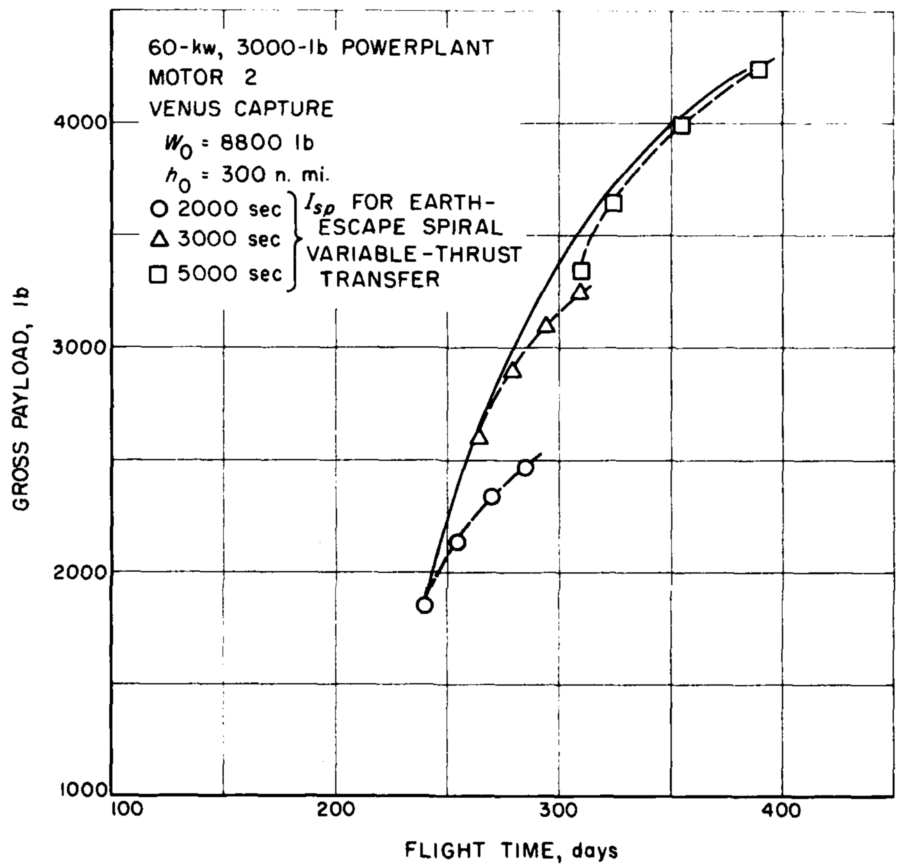

Fig. 4. Venus Capture mission, 60 kw

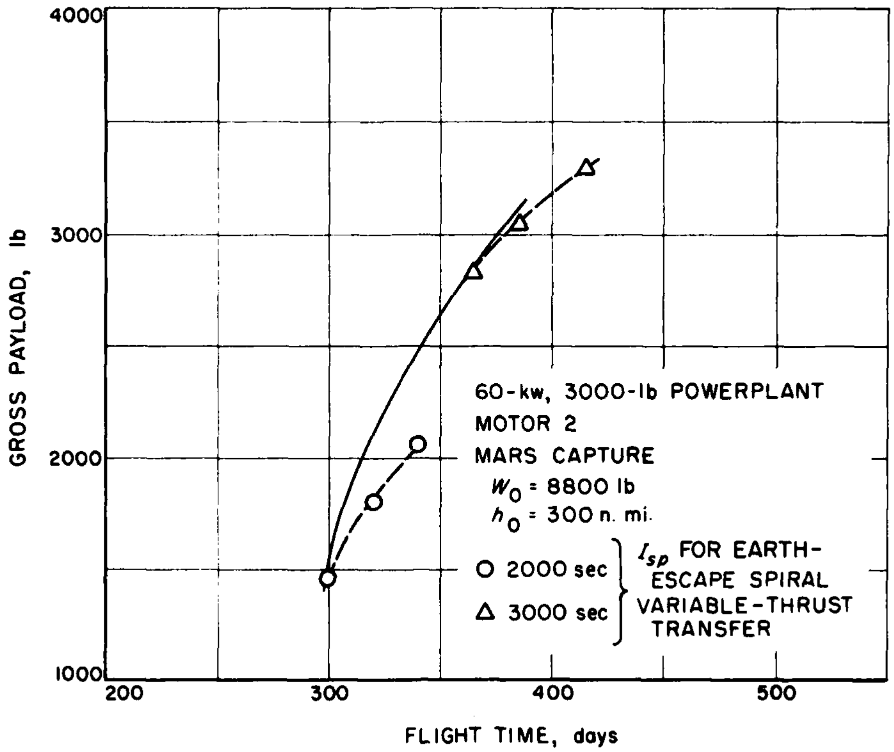

Fig. 5. Mars Capture mission, 60 kw

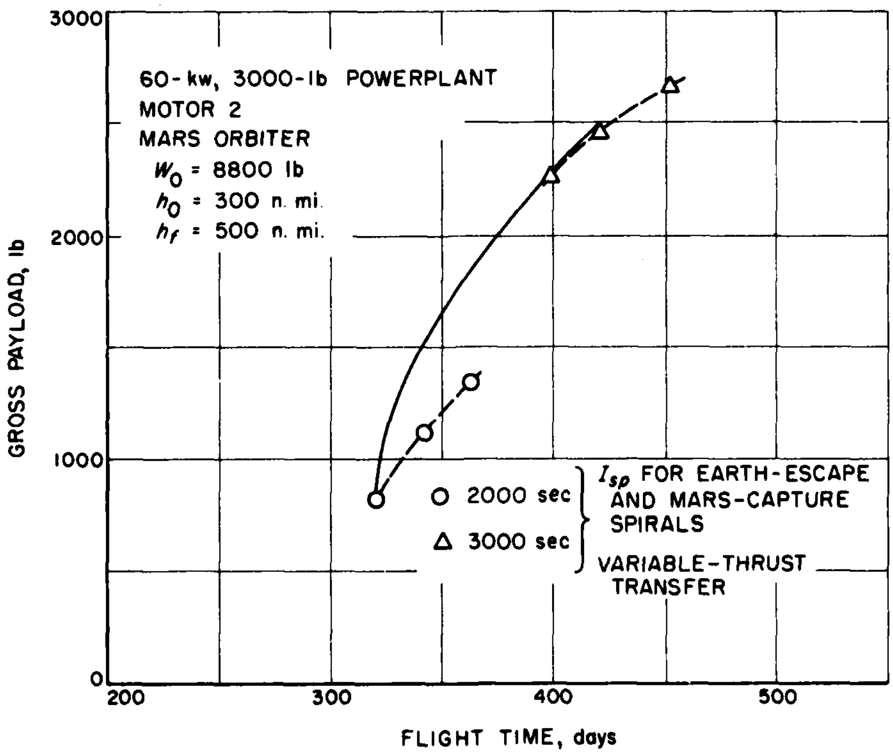

Fig. 6. Mars Orbiter mission, 60 kw

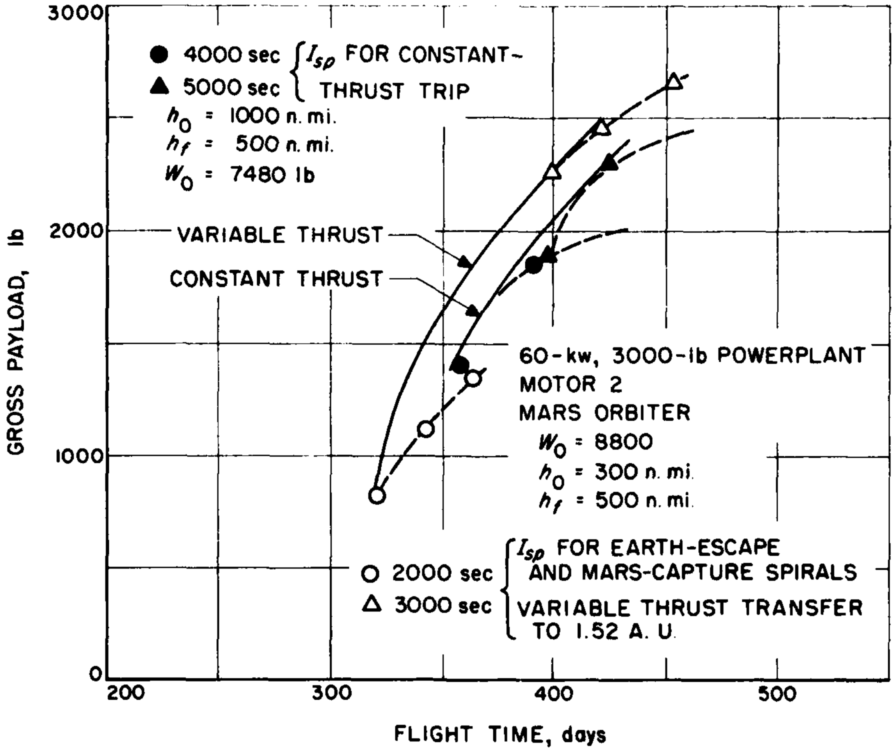

Fig. 7. Comparison of constant-thrust and variable thrust Mars Orbiter missions

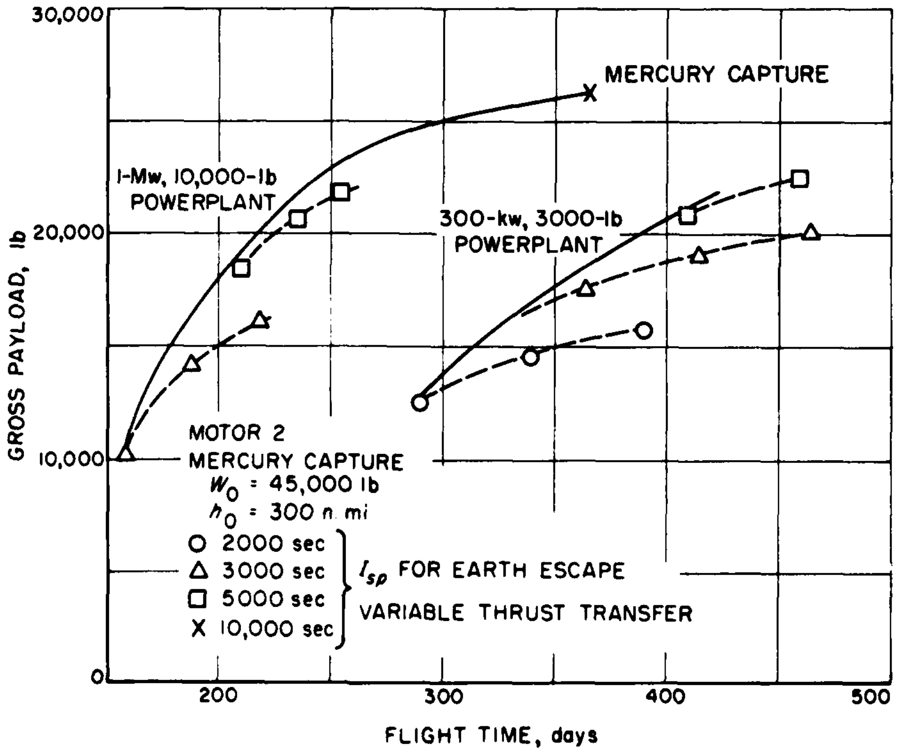

Fig. 8. Mercury Capture mission, 300 kw and 1 mw

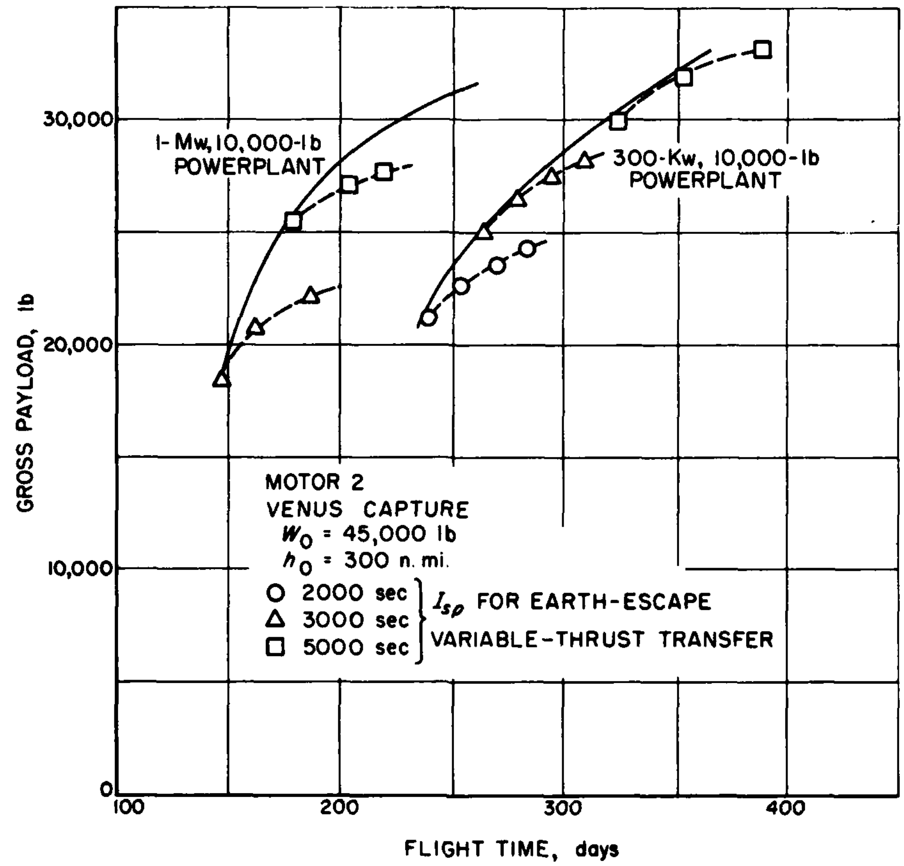

Fig. 9. Venus Capture mission, 300 kw and 1 mw

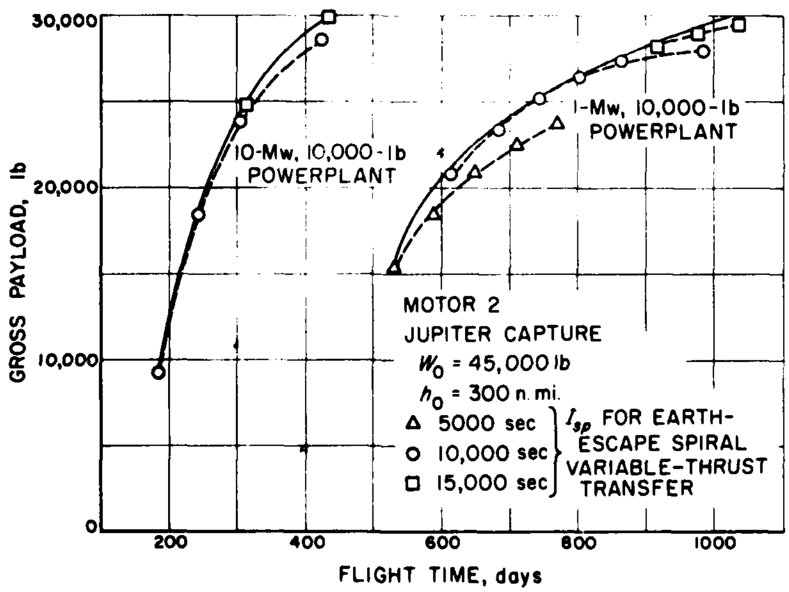

Fig. 10. Jupiter Capture mission, 1 mw and 10 mw

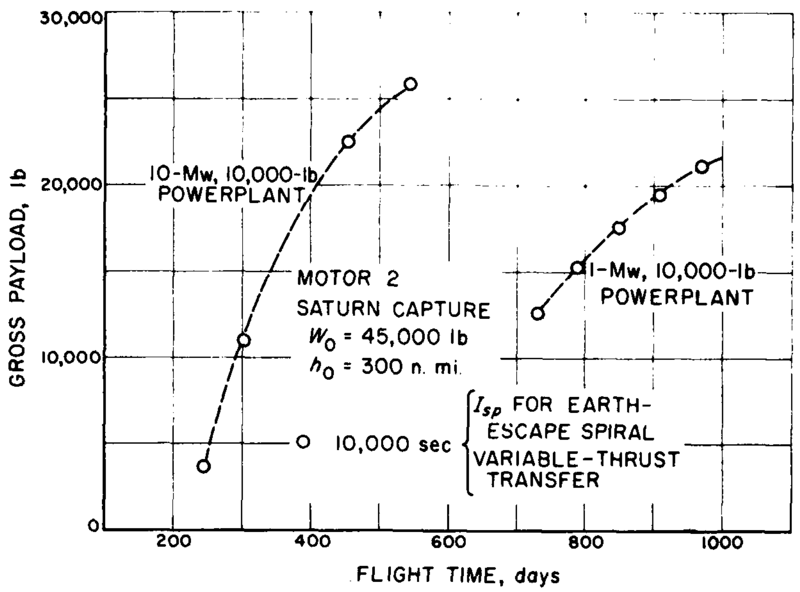

Fig. 11. Saturn Capture mission, 1 mw and 10 mw

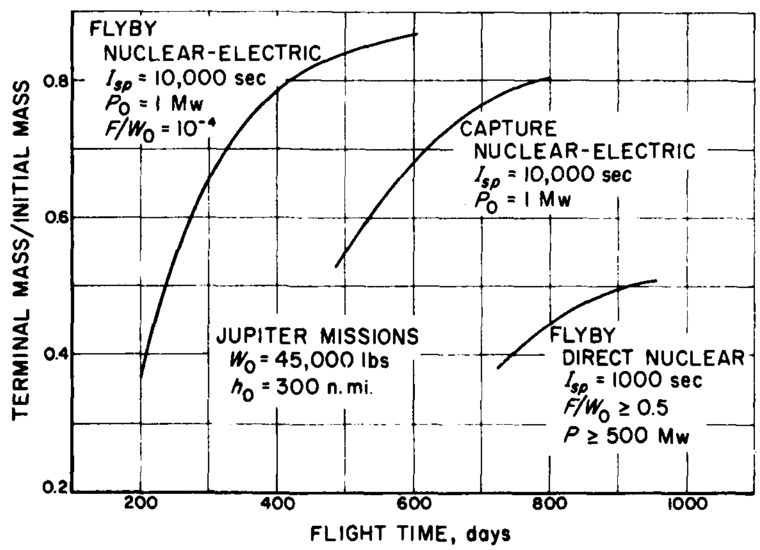

Fig. 12. Jupiter Missions comparison of nuclear and nuclear-electric propulsion

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